Orbit Control Manoeuvre Strategy for Post-Mission De-Orbiting of A Low-Earth-Orbit Satellite

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Abstract **— In this paper, we investigate a practical strategy for de-orbiting the retired satellite in low-Earth orbit for the space debris mitigation. The only means available onboard the spacecraft for performing the task is the chemical propulsion system with limited propellant provided. It is proposed to reduce the orbital perigee to reach a certain level where the atmospheric drag can play its role in lowering the satellite altitude, and eventually bringing it to re-entry within a defined period of time. The required delta-V is divided into a series under the constraints on the propulsion system and orbit control manoeuvre implementation. The results from the flight dynamics simulator suggest that a fraction of the remaining propellant available on the demonstrating mission, the Thaichote satellite, would be sufficient to accomplish the task. The strategy implementation will be another vital step in transferring the spacecraft to a safe passive state, where the fuel tank is empty, all batteries are discharged and all electronic devices are deactivated.**

Keywords **—Satellite de-orbit, Space debris, Low-Earth Orbit satellite, Thailand's Earth Observation Satellite.**

I. INTRODUCTION

Recently, the issue on space debris has been paid serious attention by the space community. The rapidly increasing number of man-made objects in operational orbit regions raises the probability of collision in space. Many missions have already experienced critical situations, and a number of orbit control manoeuvre (OCM) operations have been executed to bring the spacecraft away from the rampant debris to avoid a catastrophe that could happen to the orbiting assets, as well as the life of astronauts.

In order to reduce the risk and cost of satellite operations, most of the world-leading space organizations have introduced their standards and guidelines to mitigate space debris. The protected regions have been introduced around the Low-Earth Orbits (LEO) region, and the Geosynchronous Orbit (GEO) region as shown in Fig.1 [1]. Region A is the spherical region that extends from the Earth's surface up to an altitude (Z) of 2,000 km. Region B is a segment of the spherical shell with a lower altitude of Geostationary altitude (35,786 km) minus 200 km and an upper altitude of Geostationary altitude plus 200 km. The region is limited within $\pm 15^{\circ}$ latitude.

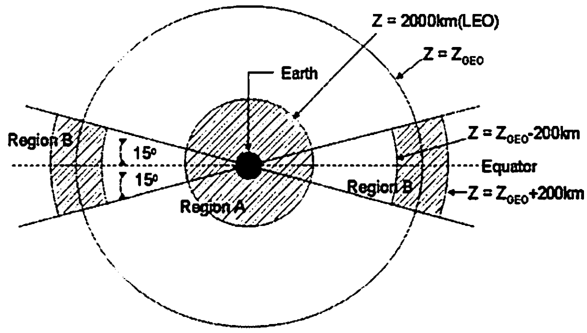

Fig. 1 Protected regions from space debris

Recently, a number of strategies have been proposed for de-orbiting obsolete spacecraft. They include sending a Satellite Re-orbiting Spacecraft (SRS) to collect the space debris [2], a tether-based de-orbiting system [3], and sail-assisted de-orbiting system [4], where a sail attached on top of a LEO spacecraft is deployed after its obsoleteness to increase the area-to-mass ratio, hence the atmospheric drag acting on the spacecraft that brings the spacecraft to natural re-entry. These strategies, however, are still during the design or conceptual verification phase. The only practical approach being used, especially for the decommissioning of satellites in GEO, is the execution of the reserved propulsion to bring the spacecraft out of the restricted area toward the so-called graveyard orbits.

In this paper, we present a strategy for de-orbiting spacecrafts in LEO using OCM. The Thaichote satellite [5] decommissioning scenario will be introduced for a demonstration of the proposed strategy. The constraints, especially on the propulsion system, are taken into account. The minimization of the propellant expenditure is of our main concern for this practical control strategy design, and its details will be described in the next section. Then, the OCM implementation, as well as the results obtained from the flight dynamics simulator will be shown and discussed. Finally, the conclusions are drawn in the last section.

II. DE-ORBITING DELTA-V STRATEGY

At an operational altitude of above 800 km, the spacecraft will experience relatively weak atmospheric drag, and without any de-orbiting strategy after the decommissioning, the spacecraft could be wandering in the LEO region for many decades or even centuries. As far as the space debris mitigation is concerned, it is recommended to de-orbit the spacecraft away from this active region. Transferring the spacecraft to an orbit outside the restricted region A shown in Fig.1, however, would not be suitable for the resource-limited missions as it requires a large amount of propellant. One practical strategy is to lower the orbital altitude as much as possible using the specially preserved fuel, and then let the atmospheric drag naturally brings the spacecraft down to re-entry.

The drag acceleration, $\vec{a}_d$, depends on the atmospheric density, which is a function of the satellite altitude, and the satellite's ballistic missile coefficient as

$$\vec{a}_d = -\frac{1}{2}\rho C_D \frac{A}{m} v_r \vec{v}_r \tag{1}$$

where, ρ is the atmospheric density, $\vec{v}_r$ is the relative velocity vector with respect to the atmosphere, A is the effective area and m is the spacecraft's mass. The coefficient of drag, C_D, is a dimensionless value which reflects the satellite susceptibility to drag forces. To meet the lifetime limit of a post-mission LEO satellite stated by the IADE and many other national guidelines of within 25 years, the spacecraft with a usual area-to-mass ratio is suggested to have an initial altitude of below 600 km [6].

As long as the minimum propellant expenditure strategy is considered, only the in-plane motion is involved. From the Gaussian form of the Variation of Parameters (VOP) [7], the dynamics of the in-plane orbital elements can be found as

$$\frac{da}{dt} = \frac{2}{n\sqrt{1-e^2}}\left\{(e\sin\nu)F_R + (1+e\cos\nu)F_S\right\} \tag{2}$$

$$\frac{de}{dt} = \frac{\sqrt{1-e^2}}{na}\left\{(\sin\nu)F_R + \left(\cos\nu + \frac{e+\cos\nu}{1+e\cos\nu}\right)F_S\right\} \tag{3}$$

$$\frac{d\omega}{dt} = \frac{\sqrt{1-e^2}}{nae}\left\{-\cos\nu F_R + \sin\nu\left(1+(e+\cos\nu)\right)F_S\right\} - \frac{\sqrt{1-e^2}\cot i \sin u}{na(1+e\cos\nu)}F_W \tag{4}$$

$$\frac{dM_0}{dt} = \frac{(1-e^2)}{nae}\left\{\left(1-\frac{e}{1-e\cos\nu}\right)F_R - \left(1+\frac{1}{1-e\cos\nu}\right)\sin\nu F_S\right\} \tag{5}$$

where a is the semi-major axis, $n \cong \sqrt{\mu/a^3}$ is the mean motion, μ is the Earth's gravitational parameter, e is the eccentricity, i is the inclination, u is the argument of latitude, ν is the true anomaly and M_0 is the initial mean anomaly. F_R, F_S and F_W are the forces acting along the radial, along-track and cross-track directions, respectively.

By using the limited propellant, the aim of our strategy is to lower the perigee as much as possible towards the region where the atmospheric drag can effectively take out the orbital energy. The perigee high, r_P, relates the semimajor axis, a, and eccentricity, e, through

$$r_P = a(1-e) \tag{6}$$

If the instantaneous change in velocity (Delta-V) as a result of a short-period propulsion execution is assumed, the change in a and e can be found as

$$\Delta a = \frac{2}{n\sqrt{1-e^2}}\left\{(e\sin\nu)\Delta V_R + (1+e\cos\nu)\Delta V_S\right\} \tag{7}$$

$$\Delta e = \frac{\sqrt{1-e^2}}{na}\left\{(\sin\nu)\Delta V_R + \left(\cos\nu + \frac{e+\cos\nu}{1+e\cos\nu}\right)\Delta V_S\right\} \tag{8}$$

where ΔV_i are impulsive changes in orbital velocity applied along each axis. It can be easily seen from Eq. (7)-(8) that the optimal delta-V firing to change both a and e is in the along-track direction. All of the ΔV firings shall be applied at the apogee to effectively lower the perigee, although the optimal position for changing the orbital size is at the perigee. In our case, the eccentricity which is not a primary parameter to be controlled will increase after each firing step, while the apogee, as well as other orbital elements remain unchanged, as depicted in Fig.2.

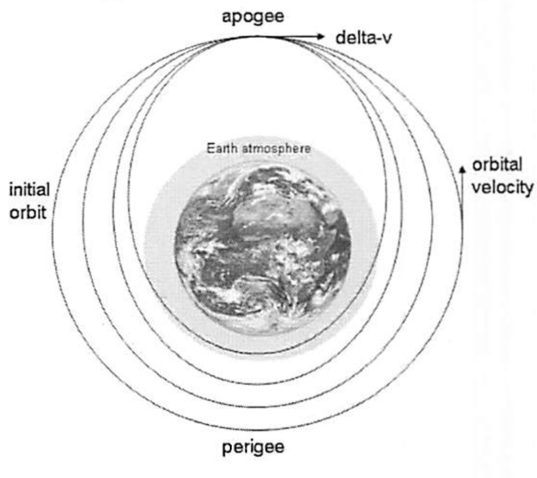

Fig. 2 Delta-V firing strategy

Finally, the changes in orbital elements as a result of each firing then become

$$\Delta a = \frac{2(1-e)}{n\sqrt{1-e^2}}\Delta V_S \tag{9}$$

$$\Delta e = -\frac{2\sqrt{1-e^2}}{na}\Delta V_S \tag{10}$$

,and

$$\Delta r_P = \frac{4(1-e)}{n\sqrt{1-e^2}}\Delta V_S \tag{11}$$

Note that, using the same delta-V, the change in perigee high is twice more effective than the change in orbital size.

III. OCM IMPLEMENTATION

The main constraint for the OCM implementation is the limitation of the available thrust. The Thaichote satellite, for example, is equipped with 4 thrusters, and they are activated through the pulse command. The maximum period for each firing, hence the number of pulses, is limited at 1000 seconds. Consequently, the maximum delta-V that can be performed at each firing is limited at about 3 m/s. For a change in orbital size larger than about 6 km, therefore, it is required to divide the delta-V into a series as depicted in Fig.2.

The monitoring of the remaining propellant after each OCM is important, and it can be evaluated from the gas pressure (*P*) and temperature (*T*) retrieved from the satellite's telemetry data. Theoretically, the required delta-V can be obtained through

$$\Delta V = \int_{t_1}^{t_2} \frac{F(t)}{m(t)} dt \tag{12}$$

where the time-varying variables *F(t)* is the thrust level, and *m(t)* is the spacecraft's mass. The equation can be represented as a function of pressure as

$$\Delta V = \int_{P_1}^{P_2} \left(\frac{F}{m}(P)\varphi'(P) \right) dP \tag{13}$$

where

$$\varphi'(P) \cong \frac{dt}{dP} = -\frac{\rho P_1 V_1}{P^2} \frac{1}{(1-c_1)q_1 + \ldots (1-c_4)q_4}$$

where P_1 and V_1 are initial pressure and volume of the propellant, respectively. c_i is the off-modulation coefficient and q_i is the propellant flow rate in each thruster. Note that *F*, *m*, and q_i are functions of *P*, whereas the propellant density, ρ, is a function of *T*. With each thruster's calibrated data provided, the final pressure, hence final mass, can be found by iteratively integrating the above equation with P_2 decreasing at each step until the required delta-V is obtained.

Note that the gas pressure, hence the thrust level, is reducing at each consecutive firing, therefore it requires longer firing duration in the later steps to gain the same delta-V. This deteriorates the OCM performance as the strategy assumes impulsive changes. Also, some other relating constraints need to be verified before the actual OCM can be performed. For instance, in the case that the OCM period is in the eclipse, it is important to check that the solar energy is sufficient for the operation, especially the attitude control task. The lightened part of the orientation operation should be sufficiently long both before and after firing.

IV. SIMULATION RESULTS

The scenario on the Thaichote satellite is introduced for the demonstration of the proposed de-orbiting strategy. The spacecraft is operating in a sun-synchronous orbit with the mean altitude of about 822 km. Its remaining onboard propellant is 45 kg. If the satellite is expected to be in service for another 10 years, the preserved propellant for the post-mission de-orbiting should not more than 27 kg. Therefore, the orbit transfer is aiming for a perigee high of 500 km.

Fig. 1 shows the simulation results obtained from the flight dynamics simulator. During the 18-day transfer period, a series of 36 delta-V firings has been applied to impulsively reduce the orbital size. There are 2 burns per day with the mean magnitude of 2.5 m/s each. The OCM verification and calibration are performed using the navigation data retrieved daily from the onboard GPS receiver. The perigee high has been lowered to 500 km as planned by using the total delta-V of 89.3 m/s which requires 26.9 kg propellant. With the same amount of propellant, the mean altitude that can be achieved by using a Hohmann-Transfer-Type strategy is about 652 km. In such minimum energy transfer strategy, the eccentricity is preserved, whereas our proposed method leaves it growing while the perigee high is shrinking as shown in Fig. 4.

After the orbit transfer process, the satellite if left orbiting in a higher drag environment. The orbit will be circularized, while the mean altitude is receding in a spiral fashion. Eventually, with its ballistic coefficient of 62 kg/m^2, the spacecraft will re-entry within 25 years as the prediction shown in Fig. 5. It is also shown that, without any de-orbit operation, the altitude barely change during such period of time.

V. CONCLUSION

We have studied a satellite's end-of-life de-orbiting. The proposed strategy effectively utilizes the limited onboard propellant to put the orbital perigee into the region where the atmospheric drag can substantially influence the satellite's orbit. Although the algorithm is simple, the implementation can be quite complicated. The satellite's motion has to be closely monitored during the orbit transfer period, in order to avoid the close approach to other space objects. The final delta-V execution is crucial, as it has to be the last one to empty the tank. The evaluation of the propulsion system near its end-of-life using the telemetry data will be less reliable. A number of ground stations, therefore, are required to specially keep the spacecraft in sight and make sure that the satellite is passivated safely.

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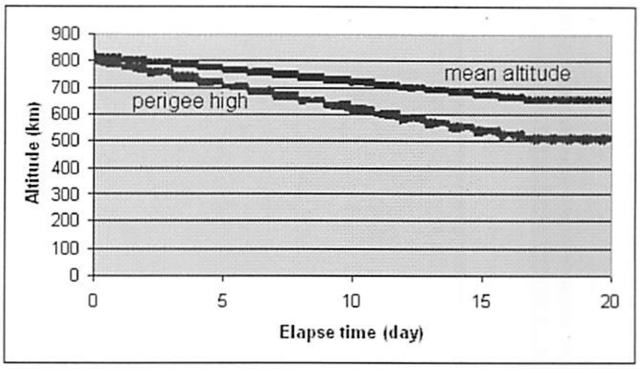

Fig. 3 Altitude profile

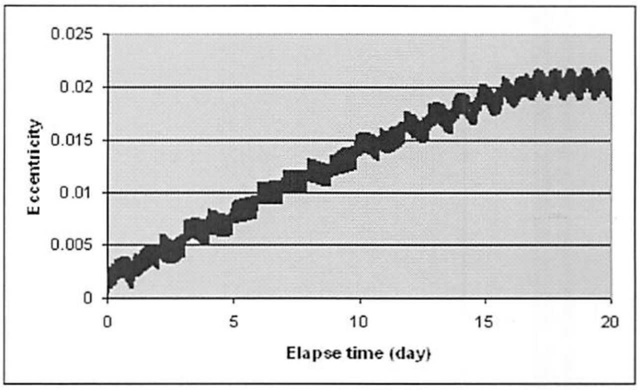

Fig. 4 Orbit eccentricity profile

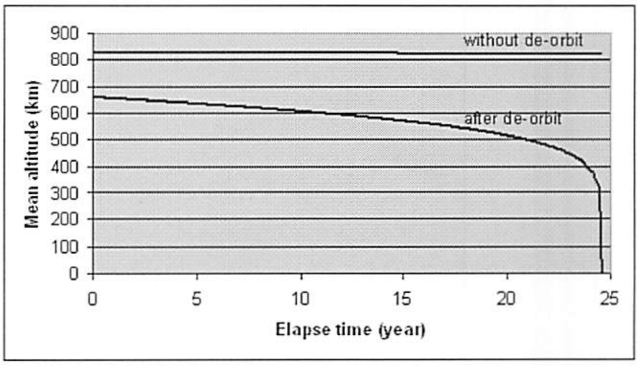

Fig. 5 Reduce of altitude toward re-entry

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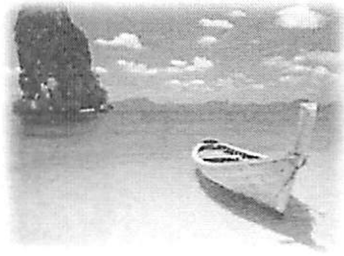


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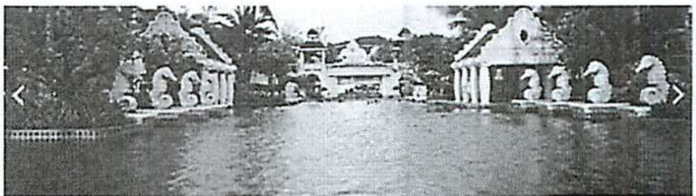

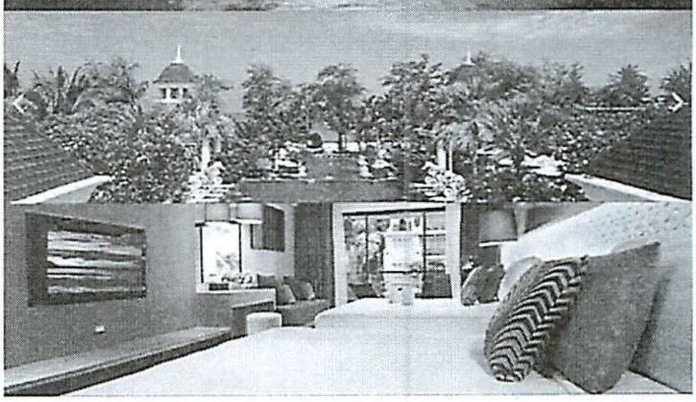

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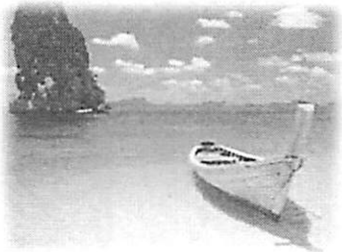

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IMPORTANT DATES

Special session proposal deadline: August 30, 2014
Special session notification: September 13, 2014
Paper submission deadline: October 13, 2014
Paper acceptance notification: November 22, 2014
Camera-ready submission deadline: December 22, 2014
Early-bird registration deadline: December 22, 2014
Conference dates: March 18-20, 2015

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The International Electrical Engineering Congress

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CONFERENCE SCHEDULE

Special session proposal deadline:	August 30, 2014
Special session notification:	~~September 13, 2014~~
Paper submission deadline:	November 3, 2014
Paper acceptance notification:	December 1, 2014
Camera-ready submission deadline:	December 22, 2014
Early-bird registration deadline:	December 22, 2014
Conference dates:	March 18-20, 2015

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เรื่อง บทความเสนอการประชุม The 2015 International Electrical Engineering Congress (iEECON2015)

อสวภ.
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เวลา 15.30 น.

เรียน อสวภ. ผ่าน รก.ผอ.สปท.

ตามที่จะมีการประชุมวิชาการ The 2015 International Electrical Engineering Congress (iEECON2015) วัตถุประสงค์ของการประชุมวิชาการนี้ เพื่อเผยแพร่งานวิจัยด้านวิศวกรรมในสาขาต่างๆ ตามเอกสารแนบ 1 โดยจัดขึ้นระหว่างวันที่ 18-20 มีนาคม 2558 ณ.จ.ภูเก็ต

ฝวว. ได้จัดทำบทความ “Orbit Control Manoeuver Strategy for Post-Mission De-orbiting of A Low-Earth-Orbit Satellite”ซึ่งเป็นบทความที่ทำการวิเคราะห์แผนการปรับวงโคจร ในกรณีปลดระวางดาวเทียม ตามเอกสารแนบ 2

จึงเรียนมาเพื่อโปรดพิจารณา หากเห็นชอบขอได้โปรดจัดส่งบทความดังกล่าวฯ ให้คณะกรรมการกลั่นกรองบทความทางวิชาการฯ เพื่อพิจารณาด้วยจะขอบพระคุณยิ่ง

(นายพรเทพ นวกิจกนก)
หน.ฝวว.

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และรูปแบบเดียวกัน

28 มค 58

เอกสารแนบ ๑

Paper Title: Orbit Control Manoeuvre Strategy for Post-Mission De-Orbiting of A Low-Earth-Orbit Satellite

Authors: Manop Aorpimai and Pornthep Navakitkanok

Comments to Authors:

By Supatcha Chaimatanan

In the framework of uncontrolled satellite de-orbiting, the authors proposed to use the OCM as a manoeuvre strategy. Instead of using the two-impulse Hohmann transfer strategy, the authors relied on a single impulse strategy to transfer the satellite to a lower-altitude co-planar orbit. The proposed methodology was tested in simulated case using parameters of the Thaichote satellite.

This is a very interesting work, and the article is easy to read.

The comments regarding the manuscript are the following:

1. In page 1, the manuscript will be easier to follow if the following information were included:
    a. A short descriptions of the Protected region;
    b. A short discussion about different types of de-orbiting options (ex. Un-controlled, controlled, semi-controlled, etc.);
    c. A short description of the tether-based de-orbiting system.

2. In the last paragraph of page 1, it should be well clarified somewhere in the paper that the propellant expenditure is 'implicitly' minimized by considering only in-plane manoeuvre, as it is not directly minimized during the computation of delta-V .
3. In page 2, left column, 2nd paragraph, a definition of 'usual' (ex: a rage of A/m ratio that is considered as 'usual') should be given.
4. The simplifications that have been made in this paper should be clearly stated somewhere.
5. In the last paragraph of page 2, a discussion about the influence of the eccentricity on the time it will take to bring the satellite to re-entry will be very interesting.
6. In page 3, to enable the interested reader to perform the simulation or to compare results obtained from other strategies, the following information should be provided:
    a. Orbit characteristics and parameters of the Thaichote satellite.
    b. The simulation platform that are used and the obtained delta-V profile.
    c. Reference to the computation of the obtained altitude (652 km) using Hohmann-Transfer-Type strategy.
7. In the conclusion section, a discussion about how to take into account the non-constant thrust direction when the impulsive thrust is applied during a period of time (instead of assume instantaneous thrust) would be very interesting.
8. The format of the references should be uniform.
9. Minor modifications, to help the reader to understand this interesting work easier, are suggested in attached manuscript.

เอกสารแนบ 2

Orbit Control Manoeuvre Strategy for Post-Mission De-Orbiting of A Low-Earth-Orbit Satellite

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a
Abstract **— In this paper, we investigate a practical strategy for de-orbiting the retired satellite in low-Earth orbit for the space debris mitigation. The only means available onboard the spacecraft for performing the task is the chemical propulsion system with limited propellant provided. It is proposed to reduce the orbital perigee to reach a certain level where the atmospheric drag can play its role in lowering the satellite altitude, and eventually ~~bringing~~ it to re-entry within a defined period of time. The required delta-V is divided into a series under the constraints on the propulsion system and orbit control manoeuvre implementation. The results from the flight dynamics simulator suggest that a fraction of the remaining propellant available on the demonstrating mission, the Thaichote satellite, would be sufficient to accomplish the task. The strategy implementation will be another vital step in transferring the spacecraft to a safe passive state, where the fuel tank is empty, all batteries are discharged and all electronic devices are deactivated.**

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Keywords **—Satellite de-orbit, Space debris, Low-Earth Orbit satellite, Thailand's Earth Observation Satellite.**

I. INTRODUCTION

Recently, the issue on space debris has been paid serious attention by the space community. The rapidly increasing number of man-made objects in operational orbit regions raises the probability of collision in space. Many missions have already experienced critical situations, and a number of orbit control manoeuvre (OCM) operations have been executed to bring the spacecraft away from the rampant debris to avoid a catastrophe that could happen to the orbiting assets, as well as the life of astronauts.

In order to reduce the risk and cost of satellite operations, most of the world-leading space organizations have introduced their standards and guidelines to mitigate space debris. ~~The~~ protected regions have been introduced around the Low-Earth Orbits (LEO) region, and the Geosynchronous Orbit (GEO) region as shown in Fig.1 [1]. Region A is the spherical region that extends from the Earth's surface up to an altitude (Z) of 2,000 km. Region B is a segment of the spherical shell with a lower altitude of Geostationary altitude (35,786 km) minus 200 km and an upper altitude of Geostationary altitude plus 200 km. The region is limited within ±15° latitude.

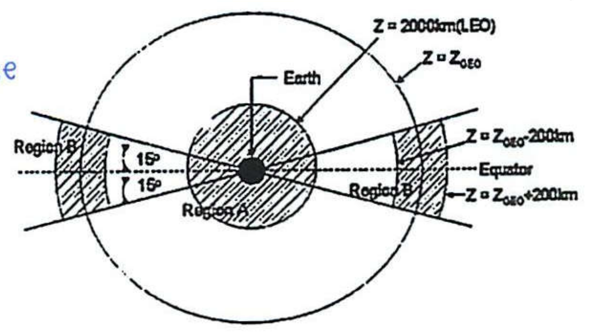

Fig. 1 Protected regions from space debris

These
Recently, a number of strategies have been proposed for de-orbiting obsolete spacecraft. ~~They~~ include sending a Satellite Re-orbiting Spacecraft (SRS) to collect the space debris [2], a tether-based de-orbiting system [3], and sail-assisted de-orbiting system [4], where a sail attached on top of a LEO spacecraft is deployed after its obsoleteness to increase the area-to-mass ratio, hence the atmospheric drag acting on the spacecraft that brings the spacecraft to natural re-entry. in the
These strategies, however, are still ~~during the~~ design or conceptual verification. ~~phase.~~ phase of The only practical approach being used, especially for the decommissioning of satellites in GEO, is the execution of the reserved propulsion to bring the spacecraft out of the restricted area toward the so-called graveyard orbits.

In this paper, we present a strategy for de-orbiting spacecrafts in LEO using OCM. The Thaichote satellite [5] decommissioning scenario will be introduced for a demonstration of the proposed strategy. The constraints, especially on the propulsion system, are taken into account. The minimization of the propellant expenditure is of our main concern for this practical control strategy design, and its Section II details will be described in ~~the next section~~. Then, the OCM implementation, as well as the results obtained from the flight dynamics simulator will be shown and discussed. Finally, the conclusions are drawn in ~~the last section~~.
Section V.
in Section III and IV respectively.

II. DE-ORBITING DELTA-V STRATEGY

At an operational altitude of above 800 km, the spacecraft will experience relatively weak atmospheric drag, ~~and~~ Without any de-orbiting strategy after the decommissioning, the spacecraft could be wandering in the LEO region for many decades or even centuries. As far as the space debris mitigation is concerned, it is recommended to de-orbit the spacecraft away from this active region. However, Transferring the spacecraft to an orbit outside the restricted region A shown in Fig.1, ~~however, would not be~~ is not suitable for the resource-limited missions as it requires a large amount of propellant. One practical strategy is to lower the orbital altitude as much as possible using the specially preserved fuel, and then let the atmospheric drag naturally brings the spacecraft down to re-entry.

The drag acceleration, $\bar{a}_d$, depends on the atmospheric density, which is a function of the satellite altitude, and the satellite's ballistic missile coefficient as

$$\bar{a}_d = -\frac{1}{2}\rho C_D \frac{A}{m} v_r \bar{v}_r, \tag{1}$$

where, ρ is the atmospheric density, $\bar{v}_r$ is the relative velocity vector with respect to the atmosphere, A is the effective area, and m is the spacecraft's mass. The coefficient of drag, C_D, is a dimensionless value which reflects the satellite susceptibility to drag forces. To meet the 25 years lifetime limit of a post-mission LEO satellite stated by the IADE and many other national guidelines, ~~of within 25 years~~, the spacecraft with a usual area-to-mass ratio is suggested to have an initial altitude of below 600 km [6].

As long as the minimum propellant expenditure strategy is considered, only the in-plane motion is involved. From the Gaussian form of the Variation of Parameters (VOP) [7], the dynamics of the in-plane orbital elements ~~can be found as~~ are given by

$$\frac{da}{dt} = \frac{2}{n\sqrt{1-e^2}}\left\{(e\sin\nu)F_R + (1+e\cos\nu)F_S\right\}, \tag{2}$$

$$\frac{de}{dt} = \frac{\sqrt{1-e^2}}{na}\left\{(\sin\nu)F_R + \left(\cos\nu + \frac{e+\cos\nu}{1+e\cos\nu}\right)F_S\right\}, \tag{3}$$

$$\frac{d\omega}{dt} = \frac{\sqrt{1-e^2}}{nae}\left\{-\cos\nu F_R + \sin\nu\left(1+(e+\cos\nu)\right)F_S\right\} - \frac{\sqrt{1-e^2}\cot i \sin u}{na(1+e\cos\nu)}F_W, \tag{4}$$

and

$$\frac{dM_0}{dt} = \frac{(1-e^2)}{nae}\left\{\left(1-\frac{e}{1-e\cos\nu}\right)F_R - \left(1+\frac{1}{1-e\cos\nu}\right)\sin\nu F_S\right\}, \tag{5}$$

where a is the semi-major axis, $n \equiv \sqrt{\mu/a^3}$ is the mean motion, μ is the Earth's gravitational parameter, e is the eccentricity, i is the inclination, u is the argument of latitude, ν is the true anomaly and M_0 is the initial mean anomaly. F_R, F_S and F_W are the forces acting along the radial, along-track and cross-track directions, respectively.

By using the limited propellant, the aim of our strategy is to lower the perigee as much as possible towards the region where the atmospheric drag can effectively take out the orbital energy. The perigee ~~high~~ height, r_p, relates the semimajor axis, a, with ~~and~~ eccentricity, e, through

$$r_p = a(1-e). \tag{6}$$

If the instantaneous change in velocity (Delta-V) as a result of a short-period propulsion execution is assumed, the change in a and e can be ~~found as~~ computed from

$$\Delta a = \frac{2}{n\sqrt{1-e^2}}\left\{(e\sin\nu)\Delta V_R + (1+e\cos\nu)\Delta V_S\right\}, \tag{7}$$

$$\Delta e = \frac{\sqrt{1-e^2}}{na}\left\{(\sin\nu)\Delta V_R + \left(\cos\nu + \frac{e+\cos\nu}{1+e\cos\nu}\right)\Delta V_S\right\}, \tag{8}$$

where ΔV_i are impulsive changes in orbital velocity applied along each axis. It can be easily seen from Eq. (7)-(8) that the optimal delta-V firing to change both a and e is in the along-track direction. All of the ΔV firings shall be applied at the apogee to effectively lower the perigee. ~~although the optimal position for changing the orbital size is at the perigee.~~ redundant In our case, the eccentricity which is not a primary parameter to be controlled, will increase after each firing step, while the apogee, as well as other orbital elements remain unchanged, as depicted in Fig.2.

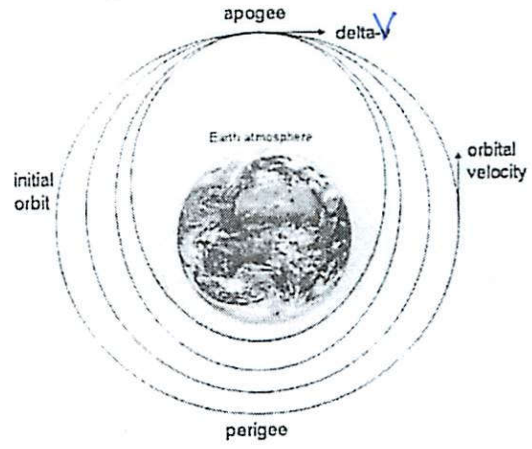

Fig. 2 Delta-V firing strategy

The authors shall consider using other notation to avoid confusion with i in Page 3.

Finally, the changes in orbital elements as a result of each firing ~~then~~ become

$$\Delta a = \frac{2(1-e)}{n\sqrt{1-e^2}}\Delta V_S, \tag{9}$$

$$\Delta e = -\frac{2\sqrt{1-e^2}}{na}\Delta V_S, \tag{10}$$

and

$$\Delta r_P = \frac{4(1-e)}{n\sqrt{1-e^2}}\Delta V_S. \tag{11}$$

Note that, using the same delta-V, the change in perigee ~~high~~ height is twice more effective than the change in orbital size.

III. OCM IMPLEMENTATION

The main constraint for the OCM implementation is the limitation of the available thrust. The Thaichote satellite, for example, is equipped with 4 thrusters, and they are activated through the pulse command. The maximum ~~period~~ duration for each firing, hence the number of pulses, is limited at 1000 seconds. Consequently, the maximum delta-V that can be performed at each firing is limited at about 3 m/s. For a change in orbital size larger than about 6 km, therefore, it is required to divide the delta-V into a series as depicted in Fig.2.

that defines the possible number of pulse?

The monitoring of the remaining propellant after each OCM is important, and it can be evaluated from the gas pressure (*P*) and temperature (*T*) retrieved from the satellite's telemetry data. Theoretically, the required delta-V can be obtained through

$$\Delta V = \int_{t_1}^{t_2} \frac{F(t)}{m(t)} dt, \tag{12}$$

where the time-varying variables *F(t)* is the thrust level, and *m(t)* is the spacecraft's mass. The equation can be represented as a function of pressure as

$$\Delta V = \int_{P_1}^{P_2} \left(\frac{F}{m}(P)\varphi'(P)\right) dP, \tag{13}$$

where

$$\varphi'(P) \cong \frac{dt}{dP} = -\frac{\rho P_1 V_1}{P^2}\frac{1}{(1-c_1)q_1 + \ldots (1-c_4)q_4},$$

where P_1 and V_1 are initial pressure and volume of the propellant, respectively. c_i is the off-modulation coefficient and q_i is the propellant flow rate in each thruster. Note that *F*, *m*, and q_i are functions of *P*, whereas the propellant density, ρ, is a function of *T*. With each thruster's calibrated data provided, the final pressure, hence final mass, can be found by iteratively integrating the above equation with P_2 decreasing at each step until the required delta-V is obtained.

Note that the gas pressure, P hence the thrust level, is reducing at each consecutive firing, therefore it requires longer firing duration in the later steps to gain the same delta-V. This deteriorates the OCM performance as the strategy assumes impulsive changes. Also, some other relating constraints ~~need~~ are needed to be verified before the actual OCM can be performed. For instance, in the case that the OCM period is in the eclipse, it is important to check that the solar energy is sufficient for the operation, especially the attitude control task. The lightened part of the orientation operation should be sufficiently long both before and after firing.

IV. SIMULATION RESULTS

The scenario on the Thaichote satellite is introduced for the demonstration of the proposed de-orbiting strategy. The spacecraft is operating in a sun-synchronous orbit with the mean altitude of about 822 km. Its remaining onboard propellant, as of February 2015, is 45 kg. If the satellite is expected to be in service, predicted for another 10 years, the preserved propellant for the post-mission de-orbiting should not more than 27 kg. Therefore, the orbit transfer is aiming for a perigee high of 500 km.

Fig.3 shows the simulation results obtained from the flight dynamics simulator. During the 18-day transfer period, a series of 36 delta-V firings has been applied to impulsively reduce the orbital size. There are 2 burns per day with the mean magnitude of 2.5 m/s each. The OCM verification and calibration are performed using the navigation data retrieved daily from the onboard GPS receiver. The perigee high has been lowered to 500 km as planned by using the total delta-V of 89.3 m/s which requires 26.9 kg propellant. With the same amount of propellant, the mean altitude that can be achieved by using a Hohmann-Transfer-Type strategy is about 652 km. In such minimum energy transfer strategy, the eccentricity is preserved, whereas our proposed method leaves it growing while the perigee high is shrinking as shown in Fig. 4.

After the orbit transfer process, the satellite if left orbiting in a higher drag environment. The orbit will be circularized, while the mean altitude is receding in a spiral fashion. Eventually, with its ballistic coefficient of 62 kg/m^2, the spacecraft will re-entry within 25 years as the prediction shown in Fig. 5. It is also shown that, without any de-orbit operation, the altitude barely change during such period of time.

V. CONCLUSION

We have studied a satellite's end-of-life de-orbiting. The proposed strategy effectively utilizes the limited onboard propellant to put the orbital perigee into the region where the atmospheric drag can substantially influence the satellite's orbit. Although the algorithm is simple, the implementation can be quite complicated. The satellite's motion has to be closely monitored during the orbit transfer period, in order to avoid the close approach to other space objects. The final delta-V execution is crucial, as it has to be the last one to empty the tank. The evaluation of the propulsion system near its end-of-life using the telemetry data will be less reliable. A number of ground stations, therefore, are required to specially keep the spacecraft in sight and make sure that the satellite is passivated safely.

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format

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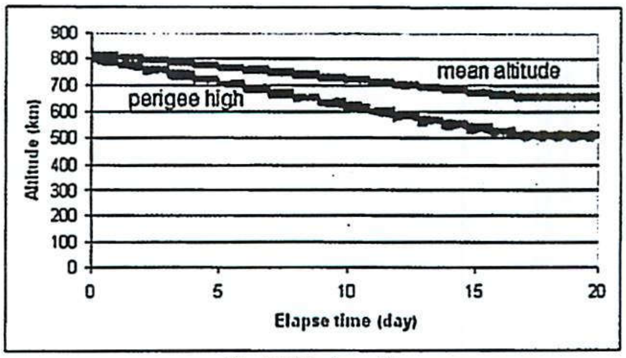

Fig. 3 Altitude profile

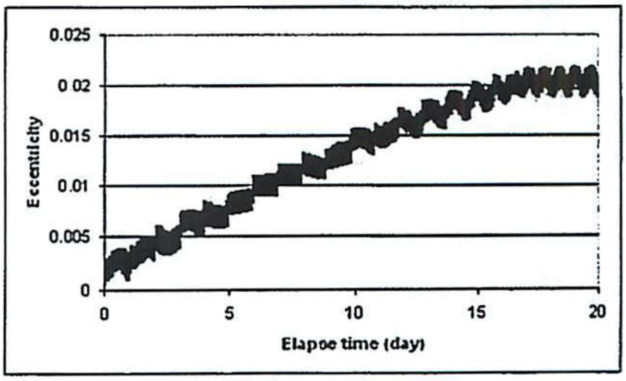

Fig. 4 Orbit eccentricity profile

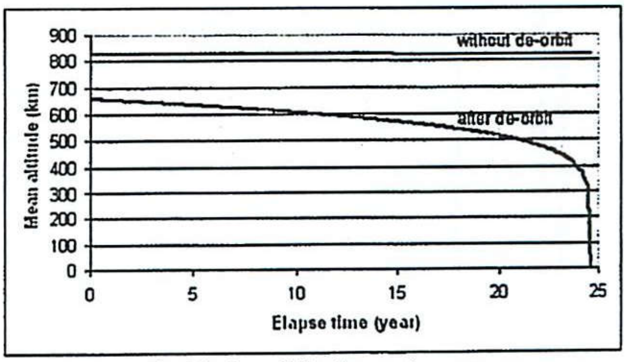

Fig. 5 Reduce of altitude toward re-entry

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เรื่อง บทความเสนอการประชุม The 2015 International Electrical Engineering Congress (iEECON2015)

เรียน อสวภ. ผ่าน รก.ผอ.สปท.

ตามที่จะมีการประชุมวิชาการ The 2015 International Electrical Engineering Congress (iEECON2015) วัตถุประสงค์ของการประชุมวิชาการนี้ เพื่อเผยแพร่งานวิจัยด้านวิศวกรรมในสาขาต่างๆ ตามเอกสารแนบ 1 โดยจัดขึ้นระหว่างวันที่ 18-20 มีนาคม 2558 ณ.จ.ภูเก็ต

ฝวว. ได้จัดทำบทความ "Orbit Control Manoeuver Strategy for Post-Mission De-orbiting of A Low-Earth-Orbit Satellite"ซึ่งเป็นบทความที่ทำการวิเคราะห์แผนการปรับวงโคจร ในกรณีปลดระวางดาวเทียม ตามเอกสารแนบ 2

จึงเรียนมาเพื่อโปรดพิจารณา หากเห็นชอบขอได้โปรดจัดส่งบทความดังกล่าวฯ ให้คณะกรรมการกลั่นกรองบทความทางวิชาการฯ เพื่อพิจารณาด้วยจะขอบพระคุณยิ่ง

(นายพรเทพ นวกิจกนก)

หน.ฝวว.

About ÆECON2015

The 2015 International Electrical Engineering Congress (iEECON2015) is a premier international academic conference organized by EEAAT, the Electrical Engineering Academic Association of Thailand. The iEECON2015 will provide a forum for researchers, engineers, and industry experts to discuss recent development, new ideas, and breakthroughs in Electrical Engineering technologies. Accepted papers will be published in Advanced Materials Research (AMR) Journal published by TTP Publications in Switzerland. AMR is indexed in Scopus. The iEECON2015 will be held in Phuket, Thailand's premier beach destination. Participants will also experience sun with the variety of entertainment and beach activities.

Conference Registration

Conference Registration
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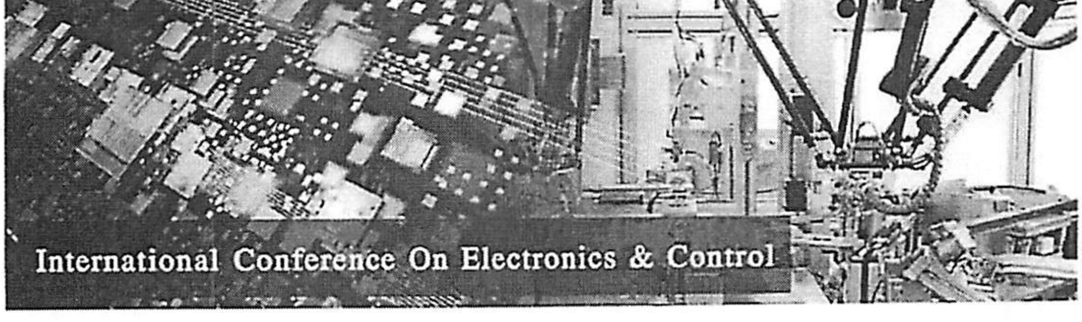
ÆECON2015 consists of five international conferences as follows :

INTERNATIONAL CONFERENCE ON COMMUNICATIONS

Communication Theory, Antennas and Propagation, Optical Communications, Microwaves, Wireless Communications, Signal Processing for Communication, Channel Coding, Multimedia Communications, Remote Sensing and Applications, Metamaterials, etc.

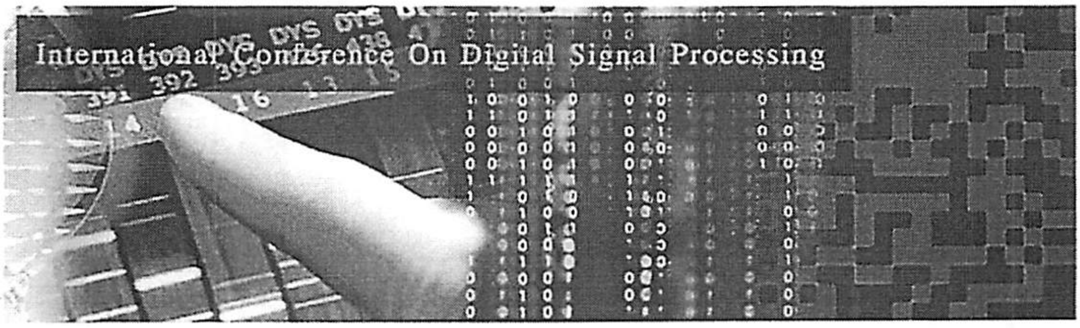
INTERNATIONAL CONFERENCE ON ELECTRONICS & CONTROL

Analog Circuits, Filters and Data Conversion, Analog and Mixed Signal Processing, Embedded Computer System, Robotics, VLSI Design, Biomedical Electronics, Industrial Electronics and automation, Adaptive Control, Electric Circuit Technology, Fault Tolerance and Detection, Semiconductor Materials, Magnetic Materials, etc.



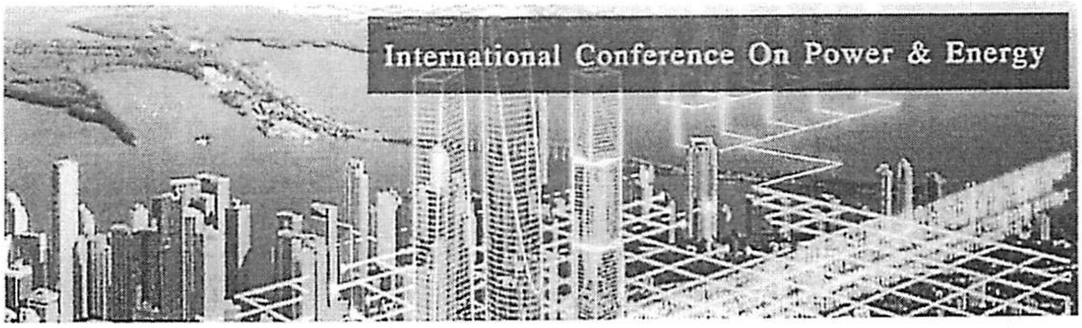
INTERNATIONAL CONFERENCE ON DIGITAL SIGNAL PROCESSING

Image and Video Processing, Audio and Speech Processing, Pattern Recognition, Biomedical Signal Processing, Computer Vision and Pattern Recognition, Adaptive Signal Processing, Machine Learning for Signal Processing, etc.



INTERNATIONAL CONFERENCE ON POWER & ENERGY

Smart Grid: Technology, Planning, Management, Operation, and Control; Electric Power Systems: Generation Transmission and Distribution, Electrical Machines, Energy Conversions, Renewable Energy Sources, Power Electronics, Energy Systems, Power Quality, High Voltage Engineering, Insulation and Materials, etc.



INTERNATIONAL CONFERENCE ON COMPUTER & IT

Computer Networks, Cloud Communication and Networking, Data Mining, Artificial Intelligence, Computational Theory, Information System, High Performance Computing, Computer Security, Software Engineering, Distributed and Parallel Computing, Web Services and Internet Computing, Multi-agent Systems, Human Computer Interaction, etc.

แบบประเมินผลการพิจารณา

ชื่อบทความ Orbit Control Manoeuver Strategy for Post-Mission De-Orbiting of A Low-Earth-Orbit Satellite

ผู้เขียน นายพรเทพ นวกิจกนก

หน่วยงาน สปท.

☑ เห็นชอบ

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ลงนาม..

(ทิพวรรณ ตันวิเศก)

วันที่ 12 กพ 58

กรุณาส่งแบบประเมินผลการพิจารณากลับมายัง ฝบว. ภายใน<u>วันที่ 13 กุมภาพันธ์ 2558</u>